8.0 OVERVIEW OF OEM METHODOLOGIES

8.1 AIRBUS INDUSTRIE

8.1.1 Probabilistic Assessment of Structure Susceptible to MSD/MED

A fatigue endurance test of a structure containing a row of nominally identical fastener holes is analogous to testing a series of simple coupons with a single fastener hole. Each single hole coupon initiates detectable cracking at different times, despite being manufactured to a common procedure; similarly, multiple hole structures will not initiate detectable cracks at the same time at each hole.

It is assumed that the crack initiation time at each site susceptible to fatigue cracking is connected to the probability distribution for fatigue endurance given by testing a large number of single hole coupons. A good estimate of the scatter (*i.e.* the standard deviation) in the fatigue endurance of details representative of the airplane structural feature is therefore fundamental to the MSD/MED assessment. The degree of variability in the manufacturing process originally used in the production of the component determines whether MSD or MED will occur, since poor quality control in manufacture results in isolated rogue flaws and the 'lead crack' scenario of traditional damage tolerance criteria. It may be extremely difficult to establish the appropriate level of scatter for a structural evaluation in an ageing airplane. Unfortunately, a supplemental fatigue endurance test programme may not furnish the required information, since 'new build' test coupons are unlikely to be representative of the original production standard, due to process and material changes over the service life of the airplane. Consequently, the conservative assumption of low scatter in fatigue endurance may have to be adopted in order to induce MSD/MED scenarios within the analysis. The assumption of high scatter suppresses multiple cracking scenarios and encourages isolated lead crack scenarios, and may result in a shorter overall fatigue endurance for a multiple hole structure.

The magnitude of the scatter directly affects the mean of the important outputs from a typical MSD fatigue assessment, *viz.* the period to first detectable crack, the period from detectable cracking to a critical crack scenario, and the overall fatigue endurance of the multiple hole structure. However, where there is any uncertainty in the scatter, a fixed standard deviation based upon the largest known values will always give a conservative analysis of fatigue endurance, although the simulation may not include many MSD/MED scenarios.

8.1.2 Calculation Procedure

- Each potential damage site in the structure (generally two per fastener hole) is allocated a different fatigue endurance, drawn randomly from the overall distribution (lognormal or Weibull) of fatigue lives for the simple coupons.

- The crack growth period is divided into intervals within a time-stepping routine, with the following calculation at each discrete time-step:
  - each damage site is checked for the initiation (or otherwise) of a fatigue crack;
  - the growth of each initiated fatigue crack is estimated through the techniques of linear elastic fracture mechanics; the stress intensity factor solutions account for the interaction of adjacent cracks and fastener holes in a simple compounding process, or through detailed finite element analysis;
  - the link-up of adjacent cracks is included within the crack growth calculation, according to the criterion of touching' crack tip plastic zones.

- The calculation stops at some pre-defined condition, *viz.* growth to a given lead crack size or structural failure according to a residual strength criterion such as the conventional crack resistance curve, or *R-curve*', techniques, with an allowance for crack interaction.

These stages form a single 'Monte Carlo' iteration; the calculation is now repeated many times, but with a different fatigue endurance (randomly allocated) at each potential damage site, such that each individual calculation represents a different damage scenario. The final output is a failure distribution (overall fatigue endurance or residual strength) associated with the multiple hole configuration. The results are generally presented graphically; for example, the overall fatigue endurance for the multiple hole configurations can be plotted against the period to the first detectable crack, as in Figure 8.1.1.

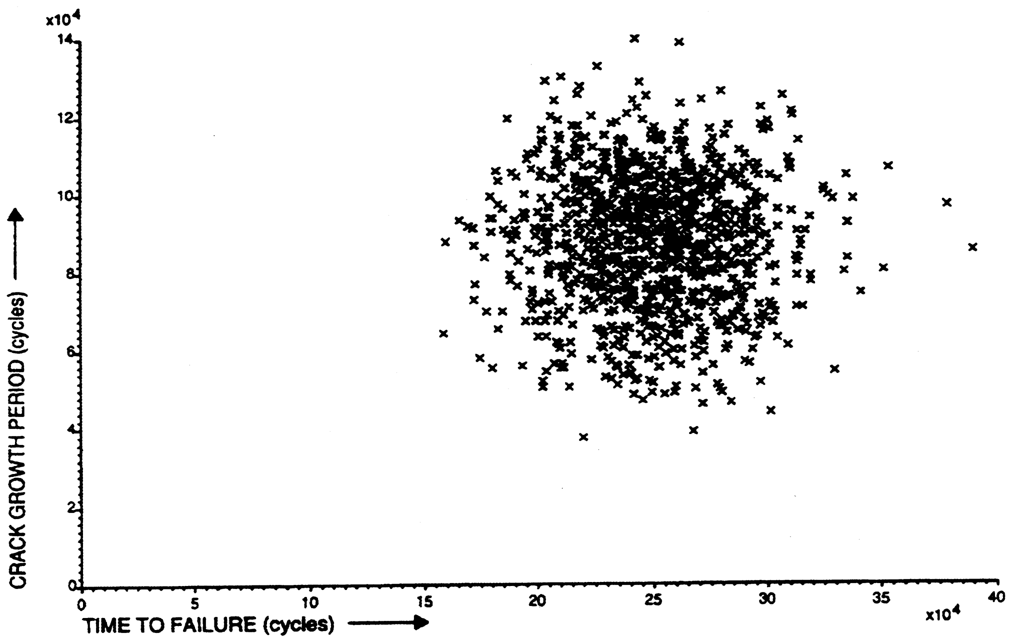


Figure 8.1.1 Fatigue endurance of multiple hole configurations.

The results can also be presented statistically by replacing the individual data points by confidence limits. The reliability of this probabilistic assessment depends on the number of scenarios considered; for example, an accuracy of 1 in 10000 requires the evaluation of at least 10000 scenarios. Figure 8.1.2 shows confidence limits on fatigue endurance for the same multiple hole configuration as in the previous illustration, along with the results of six nominally identical fatigue tests of a representative multiple hole coupon. Although the scatter in the experimental results is high, the data may be seen to be well bounded by the 99% confidence interval.

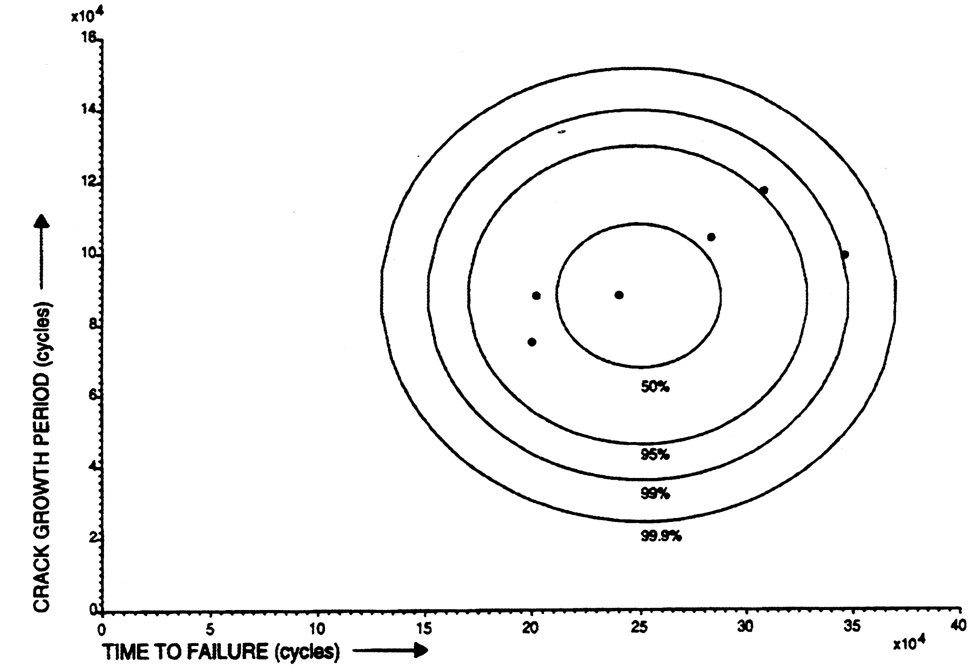


Figure 8.1.2 Confidence limits for multiple hole configurations.

8.1.3 Monitoring Period

In general, the most severe cases of adjacent multiple cracks are likely to develop only after a very long period of fatigue cycling. The most probable scenarios at earlier fatigue lives will be those associated with isolated cracks, for which a damage tolerant inspection and repair strategy should still be possible. However, the increased probability of multiple cracking in an aging airframe should be reflected within the airplane maintenance program, through the introduction of additional directed inspections providing an increased level of surveillance.

If the mean time of occurrence of failure due to WFD is established, either by calculation or test evidence, then a *'Point of WFD'* may be derived (possibly by applying a factor to the mean time for WFD) which represents a lower bound to the mean. Consequently, a *'Monitoring Period'* for operation within the MSD/MED regime may be defined, with the intention of avoiding periods where a damage tolerant inspection strategy may be inadequate because of extensive fatigue cracking. Additional inspections within the Monitoring Period are therefore initiated at some MSD/MED threshold, and continue until the Point of WFD, at which time the airframe must be modified or retired. The repeat inspection interval within the Monitoring Period will clearly be significantly shorter than for normal damage tolerance inspection programmes, in view of the increased risk of structural failure.

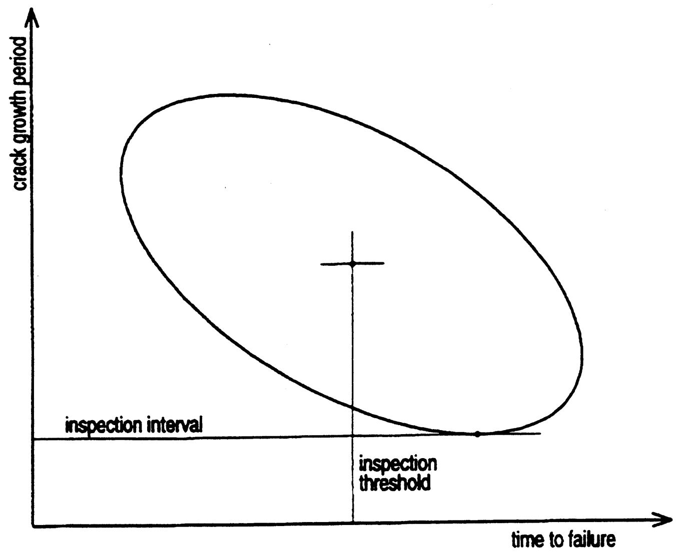


Figure 8.1.3 Inspection threshold & interval from confidence limit.

The basic parameters defining the Monitoring Period — the MSD/MED threshold, the Point of WFD, and the repeat inspection interval — may all be deduced from the results of a probabilistic assessment of structure susceptible to MSD/MED. In Figure 8.1.3, a typical confidence limit from such a calculation is shown, along with the mean time to failure from all of the different scenarios considered. The MSD/MED threshold and the Point of WFD may be established by applying appropriate factors to the mean failure period, whilst the repeat inspection interval is derived from a confidence limit on the crack growth period. A less conservative inspection interval calculation is illustrated in Figure 8.1.4, whereby the interval reduces with increasing airplane life, as a result of the reduced crack growth period in a multiple crack scenario. However, such a variable inspection programme would have to be coincident with airline maintenance schedules.

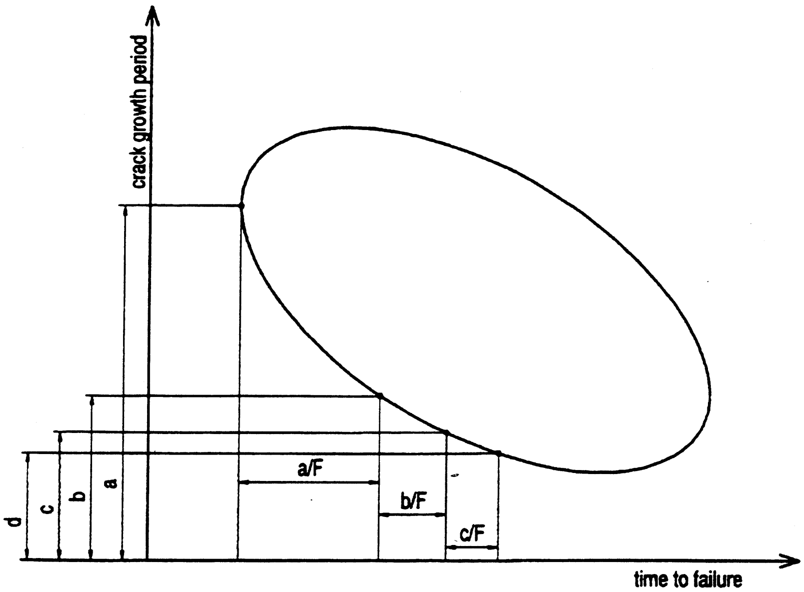


Figure 8.1.4 Modification to inspection interval.

8.2 BOEING COMMERCIAL AIRPLANES

8.2.1 Initiation / Threshold Determination

BCA currently treats MSD/MED initiation the same as MSD/MED detectable. The aim is to achieve an efficient and economical inspection program by starting it when cracks become detectable for a specified inspection method. A MSD/MED initiation with high reliability level is also achieved by focusing on very early cracking in a whole fleet. This reliability is quantifiable because the variabilities of life to cracking at different tiers of aircraft structures have been characterized by extensive testing and decades of operational fleet data. BCA uses the two-parameter Weibull probability distribution, one of the extreme value distributions,

$$F(x) = 1 - \exp\left[-\left(\frac{x}{\beta}\right)^{\alpha}\right];$$

$F(x)$ = Weibull cumulative probability function
x = fatigue life in flights
α = shape or scatter parameter
β = scale parameter or characteristic fatigue life

to model the variabilities at all different structural tiers. In general, BCA considers three structural tiers in WFD analysis, namely, critical detail, WFD component, and airplane. A critical detail, e.g., one or more adjacent rivets where early cracks will occur, is the building block of MSD/MED in a component. A WFD component, e.g., a lap splice, is an assembly of critical details. An airplane usually contains a number of underlying WFD components.

If α_1 and β_1 are the statistics of crack initiation life for critical details in a WFD component, the characteristic life β_2 of WFD component to have $r_1\%$ of critical details cracked can be estimated by, letting $x \approx \beta_2$ in the above Weibull distribution,

$$\beta_2 \approx \beta_1 \times \left[-\ln(1 - r_1\%)\right]^{-1/\alpha_1}$$

Similarly, given α_2 and β_2 the statistics of life to damage for same WFD components in an airplane, the characteristic life β_3 of airplane to have $r_2\%$ of these WFD components damaged (in $r_1\%$ of critical details) can be estimated by

$$\beta_3 \approx \beta_2 \times \left[-\ln(1 - r_2\%)\right]^{-1/\alpha_2}$$

BCA defines the MSD/MED initiation as an very early cracking event, say $r_3\%$ of airplanes in a fleet to have $r_2\%$ of WFD components damaged in $r_1\%$ of critical details, where r_1 usually is around 10 and r_2 & r_3 usually around 1. Thus, the MSD/MED initiation is estimated by the characteristic life β_4 of a fleet to have $r_3\%$ of airplanes with the prescribed damage. Let α_3 and β_3 be the statistics of life of airplane in a fleet to the prescribed damage.

MSD/MED Initiation:

$$\begin{aligned}
\beta_4 &\approx \beta_3 \times \left[-\ln(1 - r_3\%)\right]^{-1/\alpha_3} \\
&\approx \beta_1 \times \left[-\ln(1 - r_1\%)\right]^{-1/\alpha_1} \times \left[-\ln(1 - r_2\%)\right]^{-1/\alpha_2} \times \left[-\ln(1 - r_3\%)\right]^{-1/\alpha_3} \\
&\approx \beta_1 \div \prod_{i=1}^{3} \left[-\ln(1 - r_i\%)\right]^{1/\alpha_i} \\
&\approx \beta_1 \div S_{WFD}
\end{aligned}$$

S_{WFD} is a reduction factor applied to the characteristic fatigue life of critical detail to account for variabilities in all structural tiers. The characteristic fatigue life of critical detail is statistically estimated from service/test data provided data are available. Otherwise, analytical methods which involve stress calculation and in-house durability analysis procedures will be used.

The shape or scatter parameter α is estimated based on test/service data. Data over the past twenty plus years have exhibited different α s for different structural tiers. In general, scatter in critical details within a component is smaller than that between components in an airplane, and the scatter between components is smaller than that between airplanes in a fleet. That is, $\alpha_1 > \alpha_2 > \alpha_3$. The following table lists the recommended α values for pressure and externally loaded structures at different structural tiers.

	Pressure Loaded Structure	Externally Loaded Structure
Airplane	5	4
WFD Component	6	5
Critical Detail	8	6

However, different α values may be used if test/service data demonstrate otherwise.

8.2.2 Crack Growth

Crack growth analysis starts with arranging the initial MED/MSD scenario. Initial lead flaw is normally placed in the most likely or stressed detail per stress analysis results or field observation. In the case that equally stressed details exist the lead flaw will be placed in the least inspectable detail for conservatism. Secondary flaws will be placed accordingly around the lead flaw and in the adjacent details.

LEFM theory is used for calculating the growths of multiple flaws simultaneously. Specifically, the Paris law is used in the crack growth calculation with a consideration of spectrum load wherever it is necessary. Average or typical material parameters in the Paris equation are used and crack growth is deterministically calculated.

The stress intensity factors for multiple cracks growth are based on superposition of geometry factors concerning crack interaction and load redistribution. For MSD in collinear rivet holes, e.g., MSD in lap splice, BCA employs a geometry factor that was derived from full-scale lap splice panel tests. This geometry factor is made for a tip-to-tip lead crack with MSD effects considered.

However, when fractography data of actual WFD is available, the empirical crack growth curves may be used.

8.2.3 Residual Strength

BCA uses an empirical knockdown factor for residual strength when MSD is present around a lead crack. In general, it tends to give a conservative result, especially when all cracks are of similar lengths.

At present time, however, BCA only calculates Point of WFD by limiting damage growth to a conservative crack length. For MSD such as lap splice cracking without broken frames, the lead crack is limited to 1 tip-to-tip. For MED such as broken frames without skin cracks, the damage is limited to three broken adjacent frames.

8.2.4 Inspection Programs

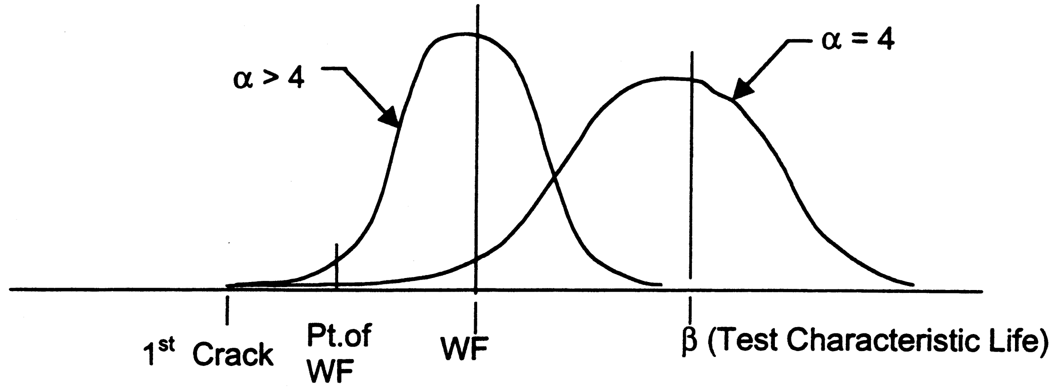
The inspection program will start at the MSD/MED initiation and end at Point of WFD. However, if there are sufficient number of airplanes inspected without evidence of WFD when the fleet leader reaches the end of program, Point of WFD may be justifiably extended.

Inspection methods and frequency will be determined based on BCA s Damage Tolerance Rating (DTR) system. This system will ensure timely detection of any MSD/MED in a fleet with a high probability of detection.

8.3 LOCKHEED-MARTIN AERONAUTICAL SYSTEMS

For the long term, LMAS plans to use available test data and the results of a limited teardown inspection of a retired L-1011 airframe to develop equivalent initial flaw size (EIFS) data. EIFS distributions would be grown forward in time using conventional crack growth methods to predict WFD (either by a Monte Carlo simulation or probability of failure calculations). There is some evidence that recent improvements in the accuracy of small crack growth predictions can produce reliable EIFS distributions, dependent only on the material fastener combination and the crack growth methodology. However, this concept has not been sufficiently validated for 2024-T3 material, and the teardown program is still in the planning stages.

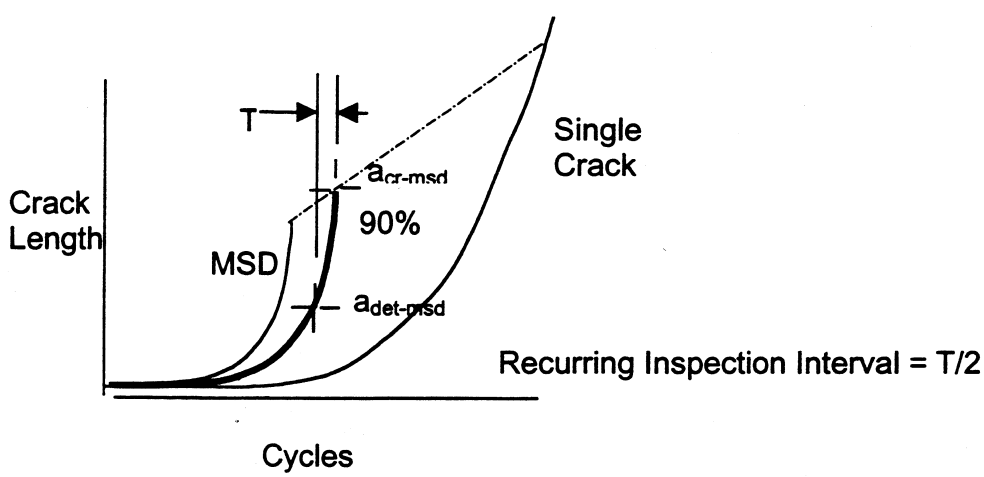
For the near term (until the EIFS concept has been validated), LMAS plans to use analysis based on the results of full scale, component, coupon tests to establish the characteristic time to crack initiation. For airplanes that have operated with stress spectra different from that applied to the test specimens (e.g., due to changes in usage), a test- demonstrated K_t will be calculated from the test results and used with the actual spectrum to estimate the fatigue life. Historical trends regarding the expected scatter in the behavior of the details will be relied upon to estimate the time to first crack and time to threshold or Point of WFD . Currently, there is thought to be some difference in the scatter of structural details within a WFD-susceptible area when compared to non-WFD details. This difference has not been quantified, but the expectation is that within a WFD location, the scatter should be less. Therefore, to be conservative in the estimation of the WFD behavior, the larger scatter factors (based, for example, on a Weibull distribution with a shape parameter, $\alpha = 4.0$) will be used to calculate the time to first crack from the characteristic life. Then, to estimate the threshold behavior, a reduced scatter ($\alpha > 4$) will be used to calculate the time from first crack to the Point of WFD, as illustrated in the following sketch.



The time to first crack is the time until there is one crack (of a detectable size) expected to exist in the WFD location.

8.3.1 Crack Growth

In a WFD scenario, with an infinite number of possible configurations of cracks growing simultaneously, there would be a different crack growth curve for each of the configurations. The differences between the crack growth curves are more pronounced as the cracks get larger due to interaction between the adjacent cracks. This, unfortunately, is also the part of the curve used to determine the recurring inspection interval. Two assumptions will represent the upper and lower bounds of the range of possible crack growth curves. As shown in the sketch below, The single crack from a loaded hole with no other active crack tips will represent the slowest growth (least conservative assumption), and the other (most conservative) extreme is when adjacent holes are cracked both sides. A Monte Carlo simulation may be the best way to consider all of the possible curves between these extremes. For the present time, however, LMAS will use an assumption that will maintain simplicity by basing the analysis on a single crack growth curve, which will be more conservative than 90% of all possible curves between the extremes, as indicated in the sketch.



The stress intensity solution for the MSD case (all holes identically cracked) is based on the superposition of correction factors for interacting cracks with the solution for cracks at both sides of a loaded hole.

8.3.2 Residual Strength

The residual strength is based on the link-up of adjacent cracks when the plastic zones touch. The Irwin equation is used to calculate the size of the plastic zones. The accuracy of predicting link-up with the Irwin equation has been shown to be dependent on the crack size and length of the ligament between the crack tips. A function is included with the Irwin model to effectively tune the link-up equation, and force agreement with the results of MSD residual strength tests across the full range of ligament lengths. At the present time, the tuning function has been developed for 2024-T3 aluminum only. Development of similar residual strength data for MSD cracks in 7075-T6 material is recommended.

8.3.3 Inspection Programs

The preliminary action will be to alert operators to areas with WFD potential and request reporting of all service findings. The notification and reporting procedures to be used will be those recommended by the AAWG and implemented by the Structures Working Group. For those areas for which a Monitoring Period is appropriate an inspection program will be developed, terminating modifications will be developed for the other areas. Lockheed may elect to develop modifications which operators may incorporate as an alternate to MSD/MED inspections.

8.4 OVERVIEW OF DELTA AIR LINES METHODOLOGY

The WFD Assessment methodology used by an STC holder may be different than the OEM s because of the lower volume of details to be analyzed. An STC holder has less incentive to develop automated analysis methods or large amounts of material data. Instead, the STC holder will generally use generic software and material data from open sources. However, this reduced volume may also allow an STC holder to use analysis methods that may be more time consuming per detail than an OEM.

The Delta Air Lines approach is a fracture mechanics based methodology, designed to be adapted to a variety of MSD/MED geometries. This approach has been used for safety management in the past for several specific cases, and compares favorable with available OEM data. We also have a large amount of service data from our large and varied fleet (approximately 600 airplanes, with 8 different models, with sub-series) to provide additional validation between our analytical models and actual events.

This methodology overview is tailored for MSD in a lap joint, but it is applicable to any geometry in which MSD is expected in collinear fastener holes.

Note: A comprehensive understanding of fracture mechanics and details of the specific geometry, coupled with fleet reliability data is required to apply this general methodology to a specific case.

8.4.1 Initiation

The Initiation calculation determines the number of cycles required for cracks to reach 0.050 in. length. This calculation is a statistical analysis, based on coupon testing of similar MSD susceptible details. The result of the coupon testing is a characteristic life of the detail.

Based on this characteristic life and an assumed scatter for Al 2024, an Initiation Table of crack initiation times is created. This table lists the cycle intervals after which new cracks will initiate. The number of fastener holes assumed present determines the confidence level of the analysis.

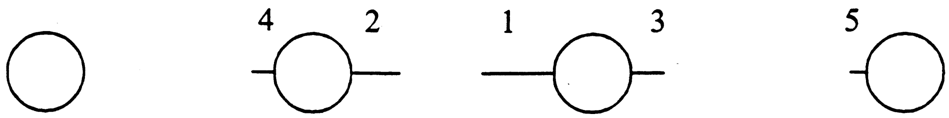
8.4.2 Crack Growth

Crack growth analysis is used to determine MSD crack lengths as a function of airplane cycle, starting from a 0.050 in. flaw. The crack growth analysis assumes a rationally conservative morphology of MSD cracks. It does not necessarily assume the worst case, but rather a cracking sequence which is conservative to some high degree of predetermined confidence.

Multiple cracks are grown using an iterative sequence of FEA models of the component. The initial model contains a single 0.050 in. crack from a hole in a high-stress location.

The succeeding model is the same, except the crack length is incremented one element longer. The stress intensity range is determined empirically by the energy released between models. Then the number of cycles required to reach the succeeding model can be calculated from da/dN[ΔK] data. Delta typically develops da/dN[ΔK] from non-proprietary sources such as Mil Handbook 5, or uses the in-house developed software incorporating Modified Forman equation and material data from NASA FLAGRO.

MSD cracks enter the model through the Initiation Table. As total cycle count reaches the next crack s initiation time in the Table, an additional 0.050 in. flaw is introduced into the model. New cracks are continually introduced as the analysis progresses. Each new crack is introduced at the worse location available, so the second crack will be an opposing crack in an adjacent hole. Generally, initiation sites continue outward from the first crack (Crack 1), as shown for five MSD cracks below.



The growth rates of all other cracks are linked to the rate of Crack 1. This linking allows cycle count between successive models to be a function of only the Crack 1 length throughout the analysis.

If MSD crack initiation occurs quickly compared to crack growth, then it is reasonable to simplify the analysis by assuming the worst case, that cracks initiate from both sides of every hole simultaneously. Under this scenario, only one hole must be modeled, with the cracked hole centered within a strip as wide as the fastener spacing. An analytical stress intensity function can be used for this strip model, instead of the FEA sequence empirical function.

8.4.3 Residual Strength

The residual strength criteria is based on the first link-up of two cracks from adjacent fastener holes. The link-up criterion is either the touching of the Irwin plastic zones or the yielding of the ligament between cracks, whichever occurs first. For the FEA empirical analysis, the plastic zones sizes and ligament stresses at limit conditions are checked at each iteration. For the strip model, ligament yield typically occurs first.

8.4.4 Inspection Threshold/Interval Determination

Inspection intervals are based on the detection of individual cracks with a 90% probability of detection, at 95% confidence. The inspection threshold is the time when a crack can first be detected, based on the crack initiation and crack growth to a detectable size. Time to initiation is based on the first cracking in the Initiation Table, factored down to account for variability among components and airplanes within a fleet.

The inspection window, from detectable to critical, is based on crack growth from detectable to the critical condition. The inspection interval is typically equal to this window divided by two, to allow two opportunities for detection.

8.5 ROUND ROBIN EXERCISES

In order to provide some insight for the regulators into the various methodologies presented in the previous section, round robin exercises were developed for the OEMs to try their methods.

Two examples were chosen for each OEM. The first is from the Boeing Company and the second from Airbus Industrie. Each example had been tested and test results were available for comparison to the OEM results. The round robins were done sequentially so that the experience gained from the first example could be applied to the second. Quantitative results are not presented here so that these examples might be used by other entities wishing to validate or confirm alternate analysis methods to their regulators.

Both examples deal with the subject of longitudinal lap splices.

8.5.1 Round-Robin Exercise Number 1

The first example, along with the requisite analysis data is shown in Figure 8.5.1. Airbus, Boeing, Lockheed Martin and Delta Air Lines all calculated the analysis parameters associated with establishing a maintenance program for MSD. All concluded that a Monitoring Period approach was valid for this particular example. Indeed all results derived were conservative with respect to the test results, however there was a significant disparity in the initial results. The AAWG then examined the reasons for the disparity. A total of nine separate areas of analysis were examined to determine where significant differences existed. In was determined that the differences in the results could be attributed to inconsistency in the use of the following parameters.

Key Parameters for MSD / MED Analysis

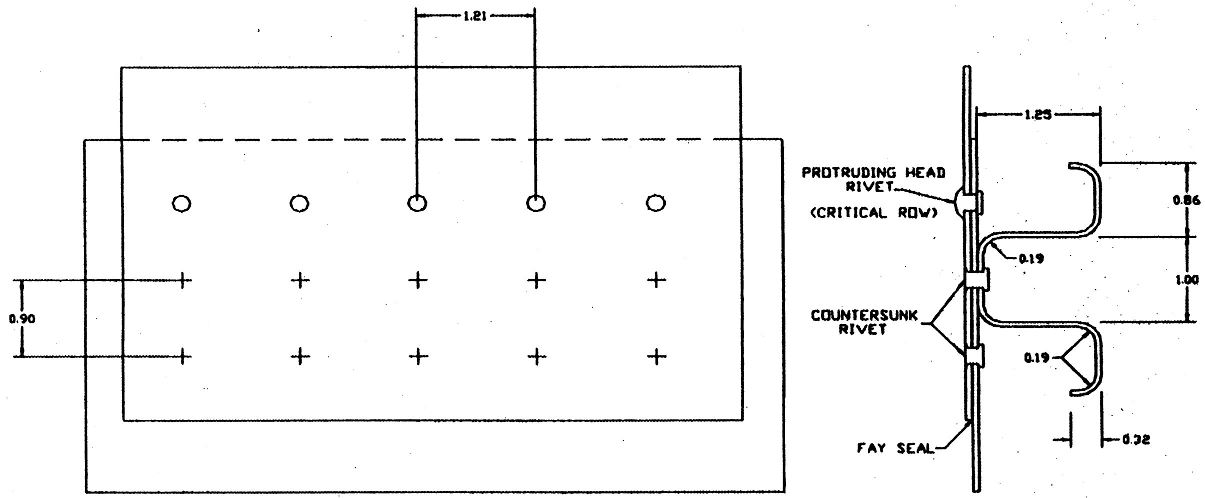
- Flaw size assumed at initiation of crack growth phase of analysis
- Material properties used (static, fatigue, fracture mechanics)
- Ligament failure criteria*
- Crack growth equations used
- Statistics used to evaluate fatigue behavior of the structure (e.g. time to crack initiation)*
- Means of determining Point of WFD*
- Detectable flaw size assumed*
- Initial distribution of flaws
- Factors used to determine lower bound behavior as opposed to mean behavior

Of the nine, the ones marked with an asterisk were considered the most significant in producing results that were different.

8.5.2 Round-Robin Exercise Number 2

The second round robin exercise, Figures 8.5.2 through 8.5.4 was conducted with the first results in mind. A set of ground rules was developed to try and minimize the disparity in the results. These ground rules were determined as shown Figure 8.5.5. In order to do some comparisons, both in-house and specified procedures were requested.

The analysis of the structural detail described in figures 8.5.2 through 8.5.4 was conducted based on coupon test results. The actual detail was tested in a full-scale test and the test results were made available to the participants after the analysis was completed. The results of round-robin number 2 showed fairly good agreement between each of the four OEMs and one airline that participated. The results were not in good comparison to the test however. Further discussion revealed that an additional factor was omitted from the analysis, that being an adjustment between coupon to full scale test. When this factor was applied to the analysis numbers reasonable answers were obtained. Figure 8.5.6 is included to show this effect in a general way. The reader is cautioned that these factors are highly dependent on design configuration, testing protocol, and other factors. A discussion of these scatter factors and mean life tendencies is detailed in section 8.5.3. Coupon to full scale test results could mean a factor on stress of as much as 1.3 or a factor on life of three. These factors have been verified through a number of manufacturer test comparisons.



SKIN
2024-T3 CLAD SHEET
t = 0.063 in
Phosphoric Acid Anodized
Fay Surface Seal

STRINGER
7075-T6 BARE SHEET
t = 0.056 in

PROTRUDING HEAD RIVET
MS20470DD
Hole Size = 0.19 1 - 0.202 in
Diameter = 0.1875 in
Head Size = 0.394 in
Bucked Head Size = 0.2625 in
2017 Aluminum
Hand Driven

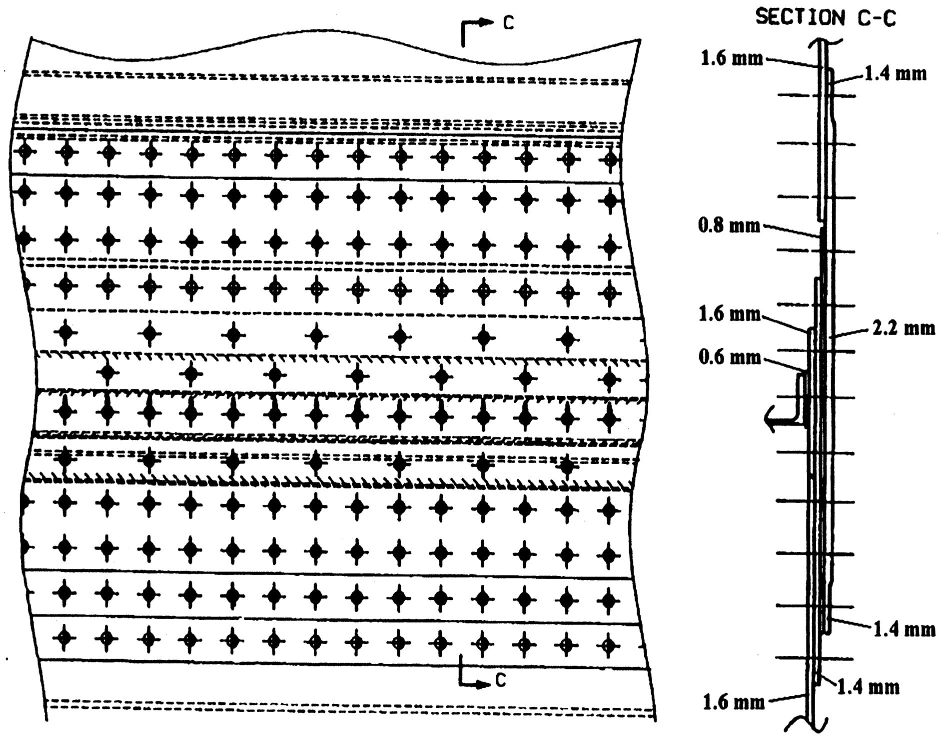
COUNTERSUNK RIVET
NAS1097
Hole Size = 0.190 - 0.196 in
Diameter = 0.1875 in
Head Size = 0.298 in
Bucked Head Size = 0.2625 in
2017 Aluminum
Hand Driven

Airplane Radius = 127 in
Frame Bay Spacing = 20 in
σ = 15 KSI (0.85*pr/t as verified by strain gage stresses at midbay due to load redistribution)
Limit Load Pressure = (Cabin Pressure + Aerodynamic Load)*1.15 = (8.9 + 0.9)*1.15 = 11.3 PSI
Limit Load Case = 0.85*11.3*127/0.063 = 19 KSI

Figure 8.5.1 — Longitudinal Lap Splice Structural Detail — Example 1

AAWG Round Robin

Example   Lap Joint Repair



skin material 2024T3
Surface treatment
chromic acid anodizing plus primer
wet assembly and wet riveting
with sealing

Rivet pitch and distance of rivet rows: 24 mm

RIVET	●	NAS1097D06
RIVET	○	NAS1097D05
RIVET	◐	NAS1097D05

Figure 8.5.2 — Lap Joint Repair — Example 2

AAWG Round Robin Exercises
Example 2 – Details

Full scale fatigue test
(fuselage radius 2820 mm)
σ_{max}= 96 MPa in 1.6 mm skin in the center between the frames, R=0 (test stress, circumferential)
limit load stress σ_{limit} = 110 MPa
limit load occurs once per life time
Characteristic life of Critical Detail:
average fatigue life of flat coupon specimens (width 160 mm) up to failure
N = 260000 cycles for σ_a = 48 MPa, R = 0.1
standard deviation: s = 0.19

<u>Skin</u>
2024 T3 clad
t=1.6 mm
Chromic acid anodized
plus primer
wet assembly and wet riveting with sealing
including faying surface
Doubler and shim material 2024T3 clad

<u>Countersunk Rivets in Lap Joint Repair</u>
NAS 1097 DD5 (solution heat treated)
Diameter: 4.0 mm
Head Size: 6.27 mm
Bucked Head Size: 5.6 - 7.5 mm
Material: Al 3.1324T31

NAS 1097 DD6 (solution heat treated)
Diameter: 4.8 mm
Head Size: 7.67 mm
Bucked Head Size: 6.7 - 8.7 mm
Material: Al 3.1324T31

The WFD evaluation is requested for the skin at the run-out of the repair doubler and shims, respectfully.

Skin stress in the center of a frame bay	100 percent
Skin stress at 1/4 length of the frame bay	97 percent
Skin stress at 3/4 length of the frame bay	97 percent
Skin stress close to the frame	89 percent

Figure 8.5.3 — Round Robin Example 2 — Analysis Data

Battelle 2024-T3 tabular data from the Damage Tolerant Design Handbook, Volume 3, page 7.5-94 compiled by UDRI for the USAF and dated December 1983. (For grain orientation: L-T, room temperature lab air environment, R-ratio = 0.0)

The two "endpoints" of this data were fit to the Paris equation to come up with the following:

da/dN = c*deltaK **n
where c=5.6153*10^{-11} and n=4.4323
da/dN = (5.6153*10^{-11})*(deltaK**4.4323)

Which yields the following tabular data points (in English units, inch & ksi):

DeltaK	da/dN
0.5	2.601*10^{-12}
4.00	2.6175*10^{-8}
16.84	1.53*10^{-5}
35.36	4.10*10^{-4}
100.0	4.111*10^{-2}

Figure 8.5.4 — Round Robin Exercise 2 — da/dN Data

Figure 8.5.5 GROUND RULES FOR AAWG-TPG ROUND ROBIN EXERCISE Number 2

The following are the general ground rules to be followed in completing the round-robin exercise.

- Airbus to provide geometry, mean life, standard deviation and other pertinent data by December 14, 1998.
- Each Participant will supply four sets of answers according to the following:

Without Fleet Variability	*With Fleet Variability*
In-house Procedures	In-house Procedures
As Specified Procedures	As Specified Procedures

- Use Mil-Handbook 2024-T3 data.
- Number of defects per airplane = 2
- Number of airplanes in fleet =50 A/P
- For specified procedure use Airbus POD curve with 6mm 95% POD
- For specified procedure assume flaw size at initiation equals 1 mm
- For specified procedure failure criterion is WFD in one frame bay.
- For specified procedure use Paris crack growth law.
- For specified procedure use $WFD_{point}=WFD_{ave}/2.0$
- For specified procedure use Inspection Start Point = $WFD_{ave}/3.0$
- Use in-house procedure for initial damage distribution.

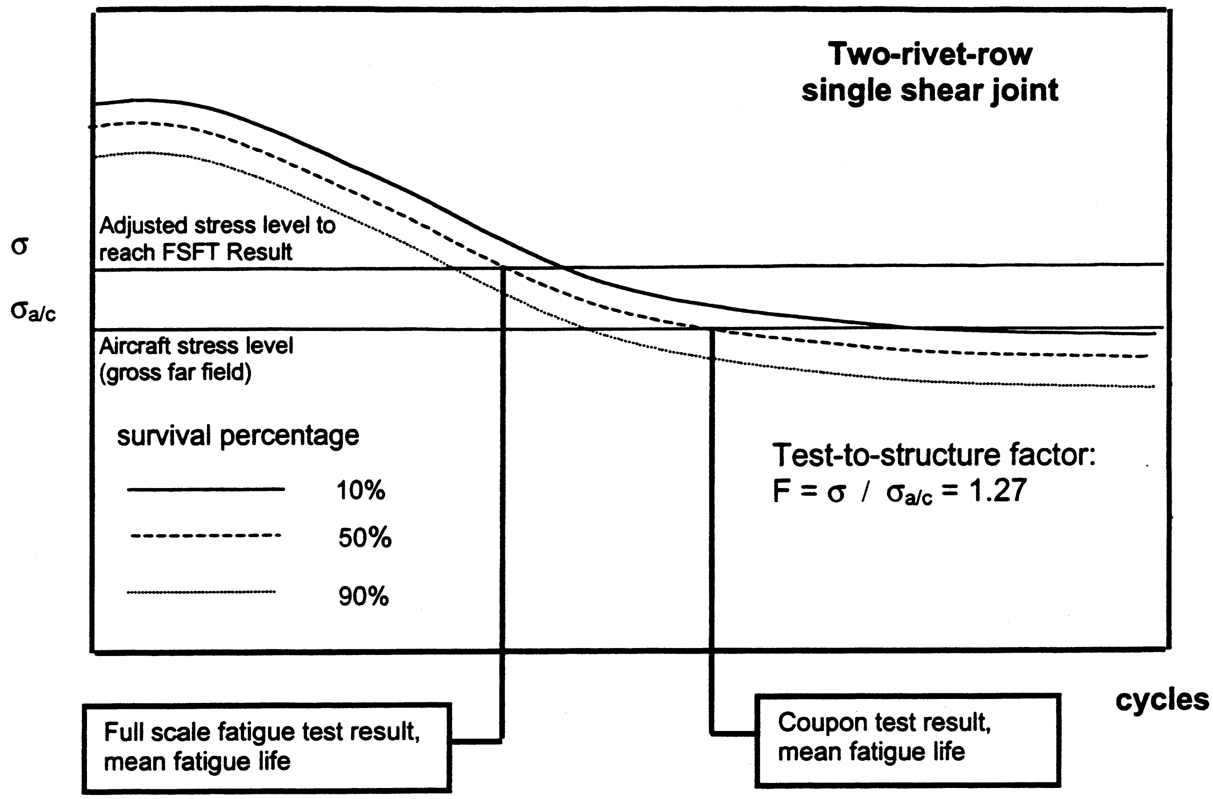


Figure 8.5.6 Typical Coupon Test —To — Full Scale Test Factor

8.5.3 Scatter Factors And Mean Life Tendencies For MSD Crack Initiation

In Appendix A of the 1993 report of the Industry Committee on Widespread Fatigue Damage, the factors to be considered when correlating test data to in-service airplanes were listed, as follows:

1. *Stress spectrum - adjustment may be accomplished using a combination of proven analysis methods and appropriate SN data or by comparative testing.*
2. *Boundary conditions - account for variations of stress levels and distributions at specific locations resulting from unrepresentative boundary conditions or load applications.*
3. *Specimen configuration effects - consideration of the effect of the number or repetitive fatigue sites in a specimen on the average initiation life and scatter band.*
4. *Material aspects - account for differences in material specification and appropriate process treatments.*
5. *Specimen geometry - conditions such as load transfer, type of fastener, secondary bending and pre-stress should represent the actual airplane configuration or be accounted for by an appropriate factor.*

6. *Environmental effects - the effects of environmental conditions should be recognized.*
7. *Scatter - scatter in test results caused by variations in specimens, test conditions and testing techniques (such as cycle rate) should be accounted for.*

Apart from item number 3, each of the considerations detailed in the above list are related to possible differences between a fatigue test specimen (either coupon, component or full-scale) and the actual behavior of the in-service airplane. The central assumption underlying the use of test evidence in predicting airplane structural fatigue is that the experimental results, usually obtained from laboratory tests on simple coupons, are representative of the airframe under service conditions. The aging airplane problem introduces additional concerns as to the validity of this assumption, such as

- For airplane types manufactured over a long period, *e.g.* more than ten years, it is likely that variations will occur in the production procedure and standard, and existing fatigue test evidence may become unrepresentative of the in-service airplane.
- Fatigue test results generated on simple coupons are unlikely to include any useful information on environmental effects such as corrosion, which are central to ensuring the continued airworthiness of the airframe.

It is generally recognized that full-scale fatigue test evidence is more accurate than the results of major component tests or coupons tests in predicting the fatigue endurance and the associated scatter factor for airframe structural components. Coupon or component test specimens are more likely than full-scale test specimens to have manufacturing processes, boundary conditions, and secondary load effects that are unrepresentative of in-service airplanes. The experimental techniques adopted during coupon or components tests, such as the environmental conditions and the cycle rate, may also be significantly different to that experienced by the airplane during operational service.

The third factor in the Industry Committee list was specifically intended to address the effect of an increase in the number of fatigue critical locations (of the same geometry and applied stress spectrum) on fatigue endurance and the associated scatter. Fatigue test results clearly show that first crack initiation occurs sooner in a group of identical repetitive details than in a single detail, provided that everything else (*e.g.* loads, specimen build standards, *etc.*) remains constant. In the case of multiple site damage and multiple element damage, the effects of load redistribution may accentuate this reduction in fatigue life.

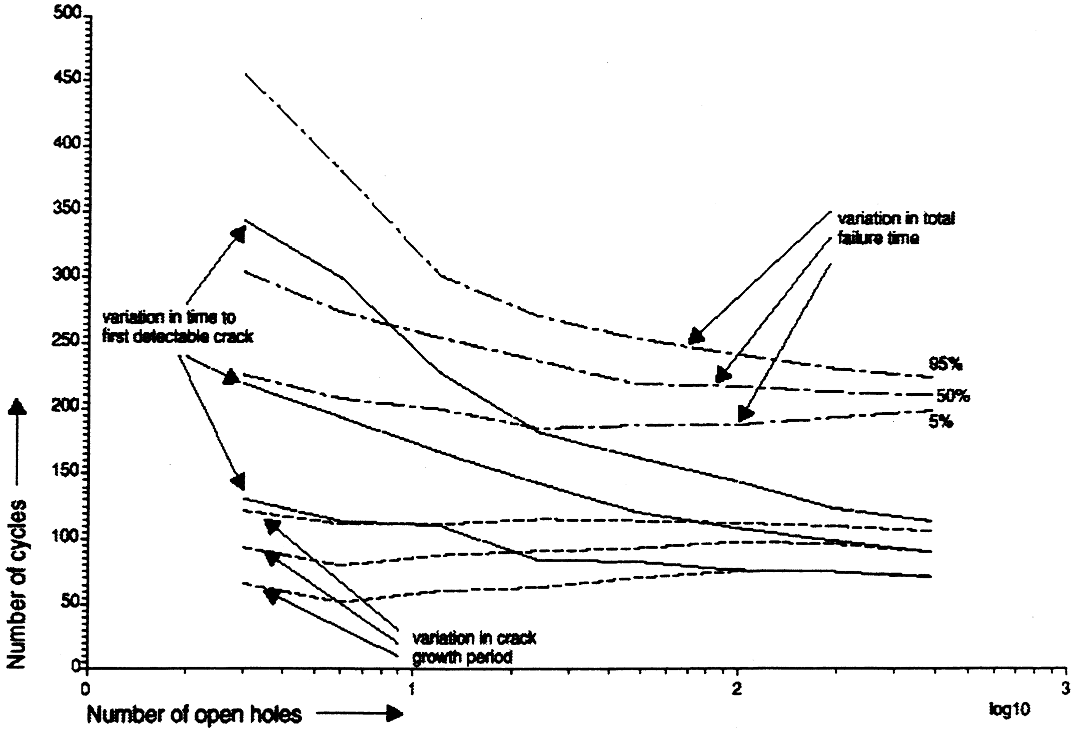


Figure 8.5.3.1 - Statistical analysis of multiple open hole specimen.

The relationship between the probability p_s of fatigue crack initiation at a single site and the probability $p_{1:n}$ of at least one such event occurring within an arbitrary number of sites n may be obtained through a simple order statistics analysis, which gives

$$p_{1:n} = 1 - (1 - p_s)^n \qquad (1)$$

This expression is independent of the nature of the probability distribution function used to model p_s (lognormal, Weibull, *etc.*). Hence, given a probability distribution function for p_s, the probability that at least one crack has developed in n potential sites (there are generally two potential sites per hole), at any specified time, may be easily obtained. The mean duration for at least one crack to initiate decreases with increasing n; the scatter in this duration (defined for example by –95% confidence limits) also decreases as n increases. An example of this behavior is shown in Figure 8.5.3.1, which gives the results of a Monte Carlo analysis of a multiple open hole specimen. A significant reduction in the mean time to the development of first detectable crack may be observed as the number of holes increases, along with a parallel reduction in the separation between the 95% confidence limits. In this example, there is not a corresponding decrease in the crack growth period between crack initiation and coupon failure.

It should be noted that the basic input is the probability of a crack initiation event occurring at a single site. If a probability distribution for initiation were defined from tests upon a simple single-hole coupon, for example, there would usually be two equally likely potential sites for crack initiation. Therefore, the distribution for p_s may be readily obtained by applying a correction to the probability distribution for the single-hole coupons, using the above expression with $n=2$. Obviously, a modification of this procedure can be applied to coupons with more than one fastener hole. A more general expression can be derived for at least m initiation events within n potential sites ($m < n$). However, the simple statistical approach breaks down in the presence of crack growth, since additional cracks are rapidly induced by load redistribution. Experience shows that the general expression can be used for $m=2$ or 3 to a reasonable accuracy. The prediction of larger numbers of newly initiated cracks requires a more representative model incorporating both the initiation and the fatigue crack growth stages.